A STUDY OF AN ADAPTIVE AIRCRAFT CONTROL SYSTEM IN A SELF-ORGANIZATION CONFIGURATION

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THESIS

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ABSTRACT

An adaptive control scheme may provide the best approach to the problem of accommodating automatic flight control systems to the variations of dynamic characteristics encountered over the flight envelope of the aircraft. The C*-Criterion for aircraft time response provides a basis for the design of such a system.

To provide a margin of safety, all control systems have redundant channels for emergency use. An adaptive control system can be designed to be self-organizing, and in such a configuration can provide its own failure monitoring, resulting in a simpler and more efficient system.

This study shows that such a system is feasible, that the response of the system is within the limits set forth in the C*-Criterion, and that the self-organizing characteristics provide reliable operation over the whole range of flight operations.

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θ - Elevation angle in Ψ θ ϕ Euler angle system.

$\dot{\theta}$ - First derivative of θ with respect to time.

$\ddot{\theta}$ - Second derivative of θ with respect to time.

α - Angle of attack

$\dot{\alpha}$ - First derivative of α with respect to time.

n - Normal acceleration of the center of gravity.

U_0 - Unperturbed aircraft velocity.

δ_e - Elevator deflection.

M_δ - Pitch acceleration per radian of δ_e

M_α - Pitch acceleration per radian of α

$M_{\dot{\alpha}}$ - Pitch acceleration per radian/sec. of $\dot{\alpha}$

M_q - Pitch acceleration per radian/sec. of q

Z_α - Normal acceleration per radian of α

Z_δ - Normal acceleration per radian of δ_e

ϵ - System error.

f - Feedback.

K_q - Fixed gain for gain changing mechanism #1.

K_n - Fixed gain for gain changing mechanism #2.

K_δ - Fixed gain for gain changing mechanism #3.

K_1 - Variable gain computed by gain changing mechanism #1.

K_2 - Variable gain computed by gain changing mechanism #2.

K_3 - Variable gain computed by gain changing mechanism #3.

l_p - Distance from center of gravity to pilot.

β_q - q coefficient of system error.

β_n - n coefficient of system error.

β_δ - C coefficient of system error.

Γ_q - Flight condition coefficient equal to $\dfrac{\beta_q - \bar{\beta}_q}{\bar{\beta}_\delta}$

Γ_n - Flight condition coefficient equal to $\dfrac{\beta_n - \bar{\beta}_n}{\bar{\beta}_\delta}$

Γ_δ - Flight condition coefficient equal to $\dfrac{\beta_\delta - \bar{\beta}_\delta}{\bar{\beta}_\delta}$

$\bar{\beta}_q$ - Aircraft coefficient equal to $M_{\dot{\alpha}} + M_q$

$\bar{\beta}_n$ - Aircraft coefficient equal to $-\dfrac{M_\alpha}{Z_\alpha} - \dfrac{M_{\dot{\alpha}}}{U_0}$

$\bar{\beta}_\delta$ - Aircraft coefficient equal to $M_\delta - Z_\delta \dfrac{M_\alpha}{Z_\alpha}$

I. INTRODUCTION

A self-organizing system may be defined as one which changes its basic structure as a function of its experiences and/or environment, with the general aim being to evolve toward some desired state of output behavior or mode of operation. Such a system can be described as having the ability to learn, and can therefore be designed to handle situations which rigidly programmed systems are not particularly adept in handling. A report by Gibson, Fu, et al., Ref. 1, gives a good introduction to, and provides background information on, learning control systems. A further and more recent introduction to the subject may be obtained from a report by Mendel, Ref. 2.

Almost all of our present aircraft use some type of air-data scheduling as the method of accommodating their automatic flight control systems to variations in the dynamic characteristics encountered over the full flight envelope of the aircraft. Such systems rely on measurements of dynamic pressure or Mach number. Also, each system must be tailored for the particular aircraft in which it will be deployed, with resulting extensive design changes from system to system, and expensive flight testing for final adjustments to insure adequate performance.

With the rapid increase in technology in the field of electronics, many new electronic instruments have appeared and proven to be

almost indispensable in the safe operation of aircraft. These instruments, already available and in use, can provide the necessary measurements for an adaptive flight control system. Such a system would depend only on measurements of dynamic performance to provide the necessary gain-programming. This should give improved system performance as compared to a programmed-gain system. Since the self-adaptive technique uses a model reference, it should be possible to make such a system almost universally applicable, requiring only a change in the model and a few parameter adjustments for each different aircraft.

The method of approach used in this study is to extend that used for the North American SIDAC controller developed by Shipley and his associates, Refs 3 and 4. This was modified by Rang, Ref. 5, from the observation that the equation for the handling qualities given by the C*-Criterion, which was developed by Tobie, Elliot, and Malcom, Ref. 6, is very similar to the basic short period equation of motion of the aircraft. A feedback system using variable gains and fixed values representative of the ranges of the aircraft coefficients over the flight envelope were used to meet the C*-Criterion requirements. The gain changing mechanism is found by a gradient technique. The system is then recast into one which should show self-organizing properties. The development of the following theory is by Shipley [Ref. 4]. It is of the basic form and shows the essential ideas. The extension to a self-organizing configuration is given in the Appendix.

II. THEORY

Beginning with the short period perturbation equations for straight and level flight, which may be written as

$$(1) \quad \ddot{\theta} = M_q \dot{\theta} + M_\alpha \alpha + M_{\dot{\alpha}} \dot{\alpha} + M_{\delta_e} \delta_e \quad ,$$

$$(2) \quad n = U_o (\dot{\theta} - \dot{\alpha}) = -Z_\alpha \alpha - Z_\delta \delta_e \quad .$$

angle of attack is eliminated because it is difficult to measure accurately and its measurement is affected by gusts and air turbulence. Equations (1) and (2) may then be written as

$$(3) \quad \ddot{\theta} = M_q \dot{\theta} + M_\alpha \left[-n/Z_\alpha - Z_\delta/Z_\alpha \, \delta_e \right] + M_{\dot{\alpha}} \left[\dot{\theta} - n/U_o \right] + M_{\delta_e} \delta_e \quad ,$$

$$(4) \quad \dot{n} = -Z_\alpha \left[\dot{\theta} - n/U_o \right] - Z_{\delta_e} \dot{\delta}_e \quad .$$

The coefficients are abbreviated as

$$\bar{\beta}_q = M_{\dot{\alpha}} + M_q \quad ,$$

$$\bar{\beta}_n = -M_\alpha / Z_\alpha - M_{\dot{\alpha}} / U_o \quad ,$$

$$\bar{\beta}_\delta = M_\delta - Z_\delta M_\alpha / Z_\alpha \quad .$$

These coefficients are negative for all flight conditions. Their values are listed in Table I. Equation (3) then becomes

$$(5) \quad -\frac{1}{\bar{\beta}_n} \ddot{\theta} + \frac{\bar{\beta}_q}{\bar{\beta}_n} \dot{\theta} + n + \frac{\bar{\beta}_\delta}{\bar{\beta}_n} \delta_e = 0$$

The C*-Criterion for aircraft response is given by

$$(6) \quad C^* = n + l_p \ddot{\theta} + U_c \dot{\theta}$$

This criterion defines the gravity forces that a pilot is subjected to as the aircraft responds to control inputs.

If C^* is required to be a multiple of the command input C_c, then

(7) $$C^* = k C_c \quad ,$$

and the criterion can be written as

(8) $$l_p \ddot{\theta} + U_c \dot{\theta} + n - k C_c = 0 \quad .$$

Equations (5) and (8) are of the same form and are exact if their coefficients are made to be equal. Thus

$$-\frac{1}{\beta_n} = l_p \quad , \quad \frac{\bar{\beta}_q}{\beta_n} = U_o \quad , \quad k C = -\frac{\bar{\beta}_s}{\beta_n} \delta_e \quad .$$

To make the problem as simple as possible, the effects of actuators, of any feedback through the aircraft control system, and of the dynamics of the control system were neglected. This ideal system is diagrammed in Figure 1. It was then assumed that the elevator command was a linear addition of the stick command and the feedback. Then

(9) $$\delta_e = c + f$$

where the feedback was a function of the gains, so that

(10) $$f = \Gamma_q \dot{\theta} + \Gamma_n n + \Gamma_s c \quad ,$$

and the gains Γ_q , Γ_n , and Γ_s were such that there was no system error. The system error was defined as

(11) $$\epsilon = \ddot{\theta} - \beta_q \dot{\theta} - \beta_n n - \beta_s c \quad .$$

When (10) was substituted into (9), and the resultant equation and (5) were together substituted into (11), the error became

$$(12) \quad \epsilon = \left[\bar{\beta}_q - \beta_q + \bar{\beta}_\delta \Gamma_q \right] \dot{\theta} + \left[\bar{\beta}_n - \beta_n + \bar{\beta}_\delta \Gamma_n \right] n + \left[\bar{\beta}_\delta - \beta_\delta + \bar{\beta}_\delta \Gamma_\delta \right] C$$

In order to hold the error to zero, the coefficients of $\dot{\theta}$, n, and C must be zero. Therefore

$$\Gamma_q = \frac{\beta_q - \bar{\beta}_q}{\bar{\beta}_\delta}, \quad \Gamma_n = \frac{\beta_n - \bar{\beta}_n}{\bar{\beta}_\delta}, \quad \Gamma_\delta = \frac{\beta_\delta - \bar{\beta}_\delta}{\bar{\beta}_\delta}$$

Values of these coefficients with

$$\beta_q = -3, \quad \beta_n = -0.2, \quad \beta_\delta = -20$$

are listed in Table II.

The feedback system was composed of three separate feedback calculators, each set for a given flight condition. These feedbacks were multiplied by gains K, and combined to give the total system feedback. The method of calculating these gains was developed by Shipley, [Ref. 4].

Letting

$$(13) \quad 2V = \frac{1}{K_q} B_q^2 + \frac{1}{K_n} B_n^2 + \frac{1}{K_\delta} B_\delta^2,$$

where

$$B_q = \bar{\beta}_q - \beta_q + \bar{\beta}_\delta \Gamma_q,$$
$$B_n = \bar{\beta}_n - \beta_n + \bar{\beta}_\delta \Gamma_n,$$
$$B_\delta = \bar{\beta}_\delta - \beta_\delta + \bar{\beta}_\delta \Gamma_\delta,$$

and K_q, K_n, and K_δ are positive numbers.

Then differentiation with respect to time gave

$$(14)\quad \frac{dV}{dt} = \bar{\beta}_\delta \left[\frac{1}{K_q} B_q \frac{d\Gamma_q}{dt} + \frac{1}{K_n} B_n \frac{d\Gamma_n}{dt} + \frac{1}{K_\delta} B_\delta \frac{d\Gamma_\delta}{dt} \right]$$

and choosing

$$\frac{d\Gamma_q}{dt} = K_q \dot{\theta} G$$
$$\frac{d\Gamma_n}{dt} = K_n n G$$
$$\frac{d\Gamma_\delta}{dt} = K_\delta c G$$

resulted in the equation

$$(15)\quad \frac{dV}{dt} = \bar{\beta}_\delta \epsilon G$$

Since $\bar{\beta}_\delta < 0$, G was chosen so that

$$G = \operatorname{sgn} \epsilon = \begin{cases} +1, & 0 \leq \epsilon_0 < \epsilon \\ 0, & -\epsilon_0 \leq \epsilon \leq \epsilon_0 \\ -1, & \epsilon < -\epsilon_0 \leq 0 \end{cases}$$

which insured that $\frac{dV}{dt}$ was negative. Therefore V will always decrease toward zero with time and the gains tended toward an ideal value for each flight condition. In this system the gains are calculated from the equations

$$\Gamma_q = K_q \int \dot{\theta} \operatorname{sgn} \epsilon \, dt$$
$$\Gamma_n = K_n \int n \operatorname{sgn} \epsilon \, dt$$
$$\Gamma_\delta = K_\delta \int c \operatorname{sgn} \epsilon \, dt$$

with

$$K_q = 5, \quad K_n = 5 \times 10^{-5}, \quad K_\delta = 500, \quad \epsilon_0 = .02 \quad .$$

III. EXPERIMENTAL PROCEDURES

The system was set up as shown in Figures 2 and 3, using the COMCOR 5000 analog computer. The basic aircraft analog was tested by making plots of $\dot{\theta}$ and n as a step input δ_e was applied to the system. The values obtained were compared with computed values of steady state amplitude, frequency, and damping ratio. Transfer functions were derived for $\dot{\theta}$ and n with respect to a step input, and the final value theorem was applied to obtain the steady state ratios.

After the basic aircraft analog was tested, the feedback system was incorporated. Each of the three feedback elements was chosen to represent a particular flight condition. The three flight conditions chosen were 0002, 0009, and 5020, where the first two numbers in each series represent the altitude in thousands of feet and the last two represent the Mach number in tenths. By choosing these particular flight conditions the operating range of the aircraft was fairly well represented. Flight condition 0002 is the low altitude, low airspeed, landing condition. Flight condition 0009 is a subsonic, low altitude condition where the aircraft has excellent response to control inputs and a high damping ratio. Flight condition 5020 is a high altitude, high speed condition where the control response is not very good and the damping ratio is low.

The three feedback elements were tested by setting the value of K for the element being tested at 1.0, and the K's for the other two elements at zero. Since the element was set up to represent exactly the flight condition being considered, the values of K should not change and the error should be zero. This proved to be the case for each of the three basic flight conditions when K was set in as an initial value. However, when operating the system with no initial values set in, the K's reached a steady state value which was somewhat different from the combination of a 1.0 and two zeros. Each flight condition did reach a steady state in which the error signal remained in its dead band. For an example, see Figures 10, 11, and 12.

After both the aircraft analog and the feedback analog were tested, the system was tested for self-organizing characteristics. For simulation purposes, the values of $\dot{\theta}$, η , and C were all taken from the aircraft analog. In a real system, these values would come from separate sensors. Sensor failures were simulated by changing the value of Γ_q , Γ_n , or Γ_s in one of the feedback elements from its calculated value to a different value. In this simulation, the values for errors were zero and one.

(11) $f = \Gamma_q \dot{\theta} + \Gamma_n \eta + \Gamma_s C$

Setting the value to zero simulated the loss of signal input from a sensor, while setting the value to one simulated an erroneous signal input. One other type of error which was simulated consisted

of using a steady input in place of one of the variable signals for $\dot{\theta}$, n , and C from the aircraft analog. This would simulate a sensor which was stuck and producing a constant output, regardless of the actual conditions. This last type error was not tested extensively, since the proper values for such signals were not known.

IV. RESULTS

The full system, for all flight conditions tested, drove its gains to values such that the system error remained in its zero range. The time required for this system adjustment varied with flight condition, frequency and amplitude of the input commands, and the number of commands given. The flight characteristics did not vary significantly during this transient time. (See Figures 13, 14, and 15)

The self-organizing characteristics of the system appear to be adequate. Simulated failures, as outlined in the method of testing, were compensated by variations in the gains. The plots in Figures 16 thru 21 show little change in aircraft flight characteristics. In some cases, the value of the error signal could not be made to remain in its dead band, but always quickly drove toward zero. For the three design flight conditions, any simulated failures outside of the design feedback element showed no effect on flight characteristics. This is as expected, since the design element should handle the flight condition by itself. When the failure was simulated in the design feedback element, there was usually a big change in the variable gains, and a slight change in performance. The flight characteristics were still satisfactory.

When multiple failures of the zero input type were simulated, the system was still able to function adequately. There was some

deterioration in the amplitude of response and the time response characteristics. The system error usually did not stay in its zero range when command signals were given, but always went rapidly to zero. The system was able to handle combinations of up to six simulated failures of inputs, as long as at least one each of the $\dot{\theta}$, η, and C inputs remained.

When multiple failures were simulated using erroneous input signals, system reaction was not as good as with the zero input failure simulation. The system was still able to reach a semi-steady state condition in which the error would drive to zero, but response was generally poorer. The poorest results were obtained when the error simulation involved two of the same quantities, such as two $\dot{\theta}$'s. In such cases, the system could not tell which input was the correct one, and the resulting performance plots were quite bad. Similar results were obtained by mixing the modes of failure together. It is considered that part of the problem encountered was the result of testing techniques, in which the inputs were increased by as much as a factor of seven to provide the erroneous signals.

One other method was used to simulate failures. A constant signal was used to replace one of the inputs. The system was able to handle this type of error very well. This type of failure was not tested extensively since representative error values were not known.

V. CONCLUSIONS

A self-organizing adaptive control scheme can handle gain scheduling within a control system in such a way that the system response is relatively invariant as required by the C*-Criterion. The ability to alter system gains as necessary to meet the response criterion gives the system better response than can be realized with a fixed gain schedule system. The system is also able to accommodate sensor failures, thus providing a built-in backup. Since the system uses a model reference, it should be easily adaptable to all types of aircraft by changing the model and the fixed feedback parameters to those of the particular aircraft.

APPENDIX

The equations of motion of the system are

$$(1)\qquad \begin{cases} \ddot{\theta} = \bar{\beta}_q \dot{\theta} + \bar{\beta}_n n + \bar{\beta}_\delta \delta_e \\ \delta_e = c + f \\ f = \Gamma_q \dot{\theta} + \Gamma_n n + \Gamma_\delta c \\ \epsilon = \ddot{\theta} - \beta_q \dot{\theta} - \beta_n n - \beta_\delta c \end{cases}$$

Changing the configuration from one with a single feedback f, which has three varying gains Γ_q, Γ_n, Γ_δ, to one which has three feedbacks of the form of f,

$$(2)\qquad \begin{cases} f_1 = \Gamma_q^1 \dot{\theta} + \Gamma_n^1 n + \Gamma_\delta^1 c \\ f_2 = \Gamma_q^2 \dot{\theta} + \Gamma_n^2 n + \Gamma_\delta^2 c \\ f_3 = \Gamma_q^3 \dot{\theta} + \Gamma_n^3 n + \Gamma_\delta^3 c \end{cases}$$

but with fixed gains, In matrix form

$$(3)\qquad \Gamma = \begin{bmatrix} \Gamma_q^1 & \Gamma_n^1 & \Gamma_\delta^1 \\ \Gamma_q^2 & \Gamma_n^2 & \Gamma_\delta^2 \\ \Gamma_q^3 & \Gamma_n^3 & \Gamma_\delta^3 \end{bmatrix}$$

is constant. The three feedbacks are combined with varying gains.

$$(4)\qquad K = \begin{bmatrix} K_1 \\ K_2 \\ K_3 \end{bmatrix}$$

to give the final form of the input into the elevator channel

(5) $\quad f = K_1 f_1 + K_2 f_2 + K_3 f_3$

To facilitate the derivation, change to vector-matrix notation.

Let

$$X = \begin{bmatrix} \dot{\theta} \\ n \\ c \end{bmatrix}, \quad \bar{\beta} = \begin{bmatrix} \bar{\beta}_q \\ \bar{\beta}_n \\ \bar{\beta}_\delta \end{bmatrix}, \quad \beta = \begin{bmatrix} \beta_q \\ \beta_n \\ \beta_\delta \end{bmatrix}, \quad \gamma = \begin{bmatrix} \Gamma_q \\ \Gamma_n \\ \Gamma_\delta \end{bmatrix}.$$

The set of equations from (1) then yields

(6) $$\begin{cases} \ddot{\theta} = \bar{\beta}' x + \bar{\beta}_\delta f = \bar{\beta}' x + \bar{\beta}_\delta \gamma' x \\ \epsilon = (\bar{\beta}' - \beta' + \bar{\beta}_\delta \gamma') x \end{cases},$$

where the prime denotes the transpose of the matrix.

Consider the function

(7) $$2V = (\bar{\beta}' - \beta' + \bar{\beta}_\delta \gamma') Q (\bar{\beta} - \beta + \bar{\beta}_\delta \gamma) ,$$

where Q is a positive definite matrix to be chosen later.

Differentiation gives

(8) $$\frac{dV}{dt} = (\bar{\beta}' - \beta' + \bar{\beta}_\delta \gamma') Q \bar{\beta}_\delta \frac{d\gamma}{dt}$$

Since V is to be driven to zero, which in turn means ϵ is driven to zero, the requirement is

$$\frac{dV}{dt} \leq 0$$

Thus, choose

(9) $$\frac{d\gamma}{dt} = Q^{-1} x G$$

so that

$$\frac{dV}{dt} = \bar{\beta}_\delta \epsilon G$$

Since the parameter $\bar{\beta}_s$ is always negative, taking

$$G = \operatorname{sgn} \epsilon$$

makes

$$\frac{dV}{dt} = \bar{\beta}_s \, |\epsilon|$$

so that

$$\frac{dV}{dt} \leqslant 0$$

If

$$(10) \qquad Q^{-1} = \begin{bmatrix} K_q & 0 & 0 \\ 0 & K_n & 0 \\ 0 & 0 & K_s \end{bmatrix}$$

then the system is the same as given in the body of the thesis.

Equation (5) combined with (2) is

$$(11) \qquad f = K' \, \Gamma \, x$$

But also from equation (1)

$$(12) \qquad f = \gamma' \, x$$

Therefore

$$\gamma = \Gamma' K$$

Equation (9) is then

$$\frac{d\gamma}{dt} = \Gamma' \frac{dK}{dt} = Q^{-1} x \, G$$

or

$$(13) \qquad \frac{dK}{dt} = (Q \Gamma')^{-1} x \, G$$

Now choose

$$(Q\,\Gamma')^{-1} = Q_o^{-1}\,\Gamma$$

so that

$$\text{(14)} \qquad \frac{dK}{dt} = Q_o^{-1} \begin{bmatrix} f_1 \\ f_2 \\ f_3 \end{bmatrix} G$$

with

$$Q_o^{-1} = \begin{bmatrix} K_q & 0 & 0 \\ 0 & K_n & 0 \\ 0 & 0 & K_s \end{bmatrix}$$

Equation (14) becomes

$$\text{(15)} \qquad \begin{cases} \dfrac{dK_1}{dt} = K_q\, f_1\, G \\[2ex] \dfrac{dK_2}{dt} = K_n\, f_2\, G \\[2ex] \dfrac{dK_3}{dt} = K_s\, f_3\, G \end{cases}$$

These produce the variable gains used in the self-organizing system. Since

$$Q = \Gamma^{-1}\, Q_o\, (\Gamma^{-1})'$$

If Γ is non-singular and Q_o is positive definite, then Q is positive definite, which is necessary for concluding that ϵ goes to zero as V goes to zero.

FLIGHT CONDITION	0002	0009	1504	3006	4509	5020
ALTITUDE	0	0	15,000	30,000	45,000	50,000
MACH NO.	0.2	0.9	0.4	0.6	0.9	2.0
U_0	223	1005	423	597	872	1938
$-M_\alpha$	1.3	38	2.93	3.6	4.35	42.4
$-M_{\dot{\alpha}}$	.26	1.38	.33	.27	.20	.04
$-M_q$	.43	2.61	.56	.51	.34	.48
$-M_\delta$	2.8	58.4	6.3	7.5	6.52	14.9
$-Z_\alpha$	83	2420	228	262	277	817
$-Z_\delta$	13.2	313	31.5	38.2	35.8	102
$-\bar{\beta}_\delta$	2.58	53.5	5.90	6.93	3.78	9.61
$-\bar{\beta}_q$	.69	4.0	.89	.78	.54	.52
$-\bar{\beta}_n$	.0146	.0143	.0121	.0133	.0158	.0519
Γ_q	90	-.019	.36	.32	.65	.26
Γ_n	.0021	.0001	.0013	.0010	.0011	.0033
Γ_δ	6.75	-.63	2.39	1.88	4.29	1.08

TABLE I AIRCRAFT PARAMETERS

(F4 AIRCRAFT Units in radians, feet, seconds)

K_q	.05
K_n	.0005
K_s	500
Γ_q^1	.895
Γ_q^2	-.0187
Γ_q^3	.258
Γ_n^1	.00213
Γ_n^2	.000107
Γ_n^3	-.00331
Γ_s^1	6.75
Γ_s^2	-.626
Γ_s^3	1.083
ϵ_0	.02
β_q	-3.0
β_n	-.02
β_s	-20

TABLE II FEEDBACK CONSTANTS

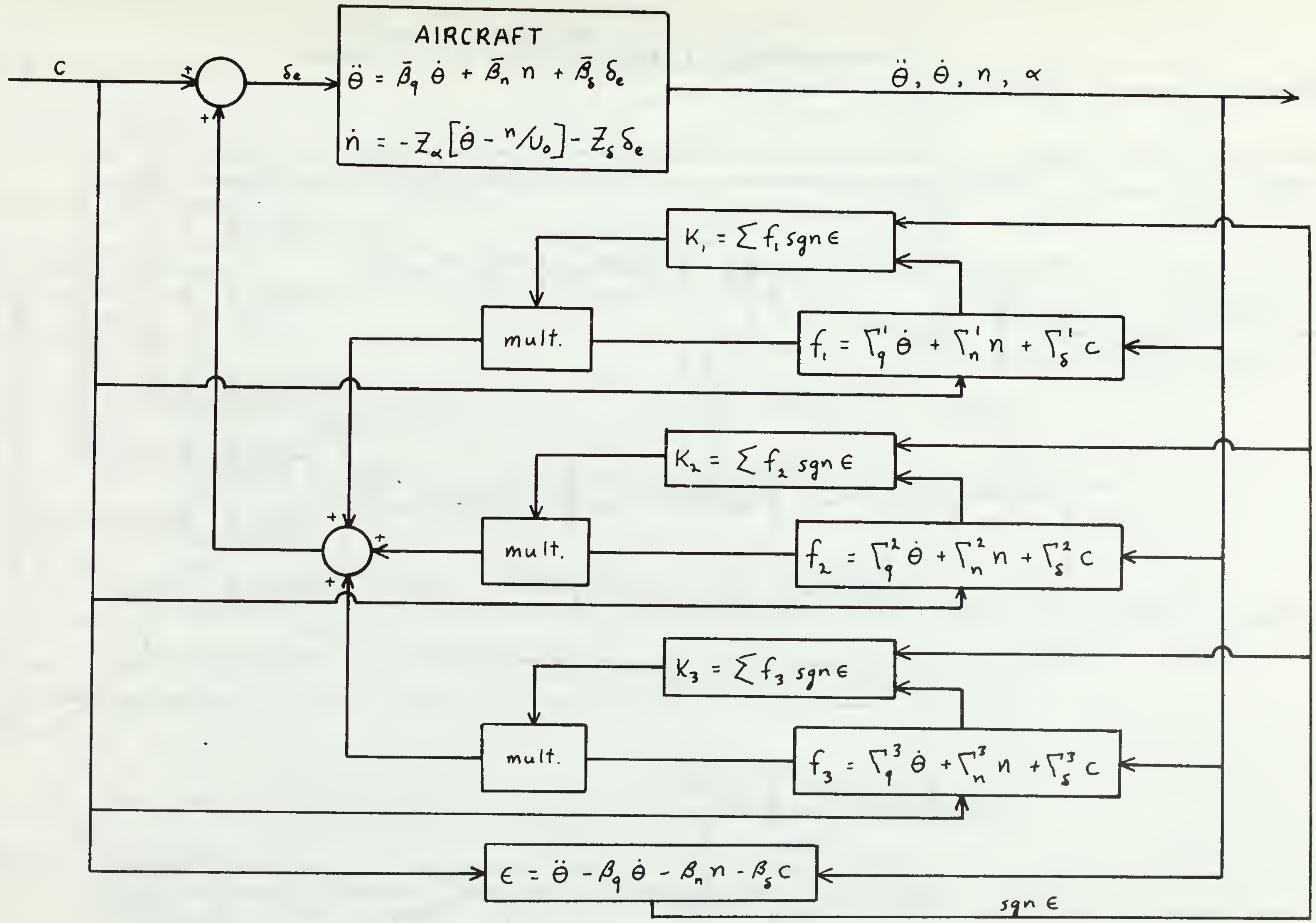

FIGURE 1 CONTROL CONFIGURATION

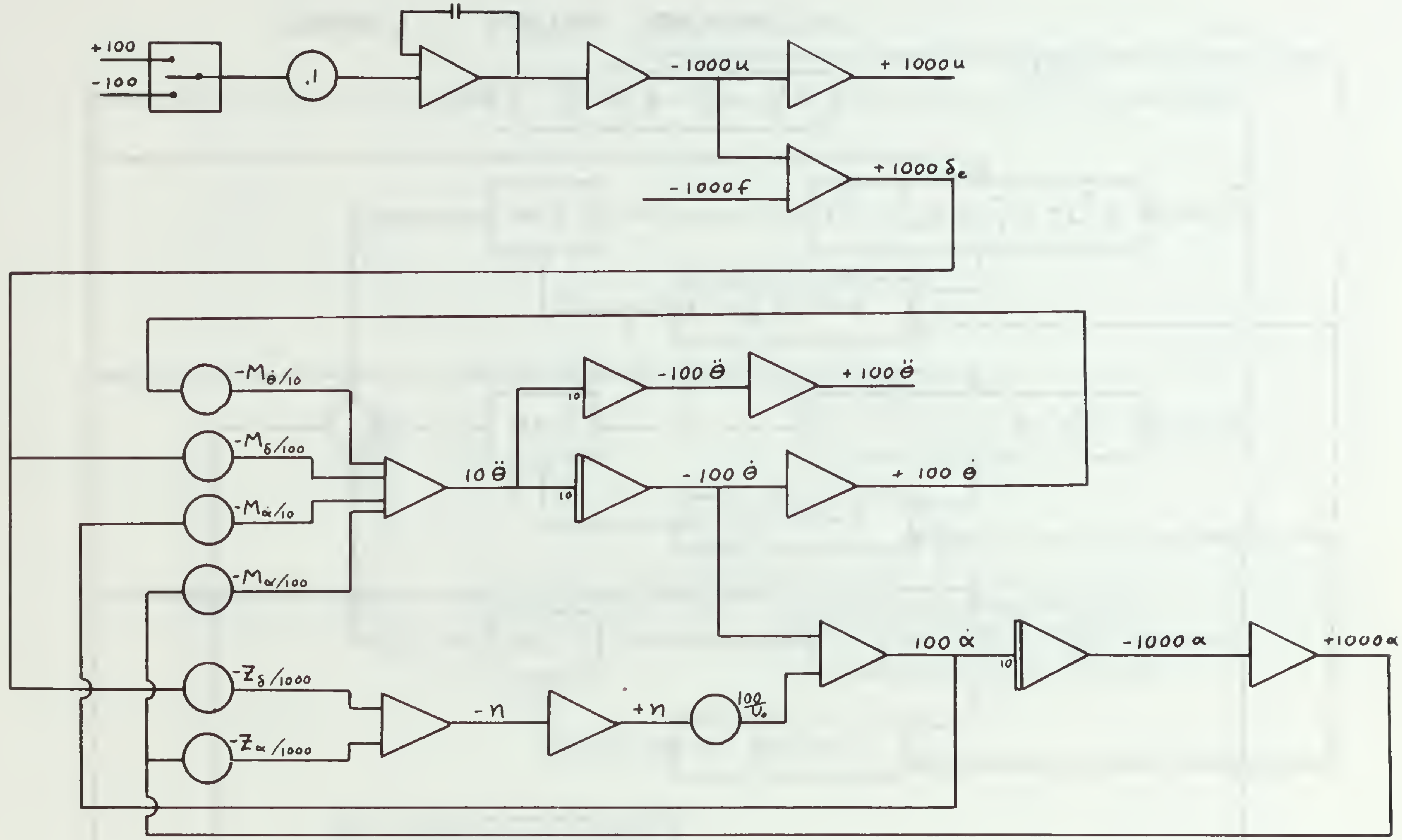

FIGURE 2 BASIC AIRCRAFT ANALOG

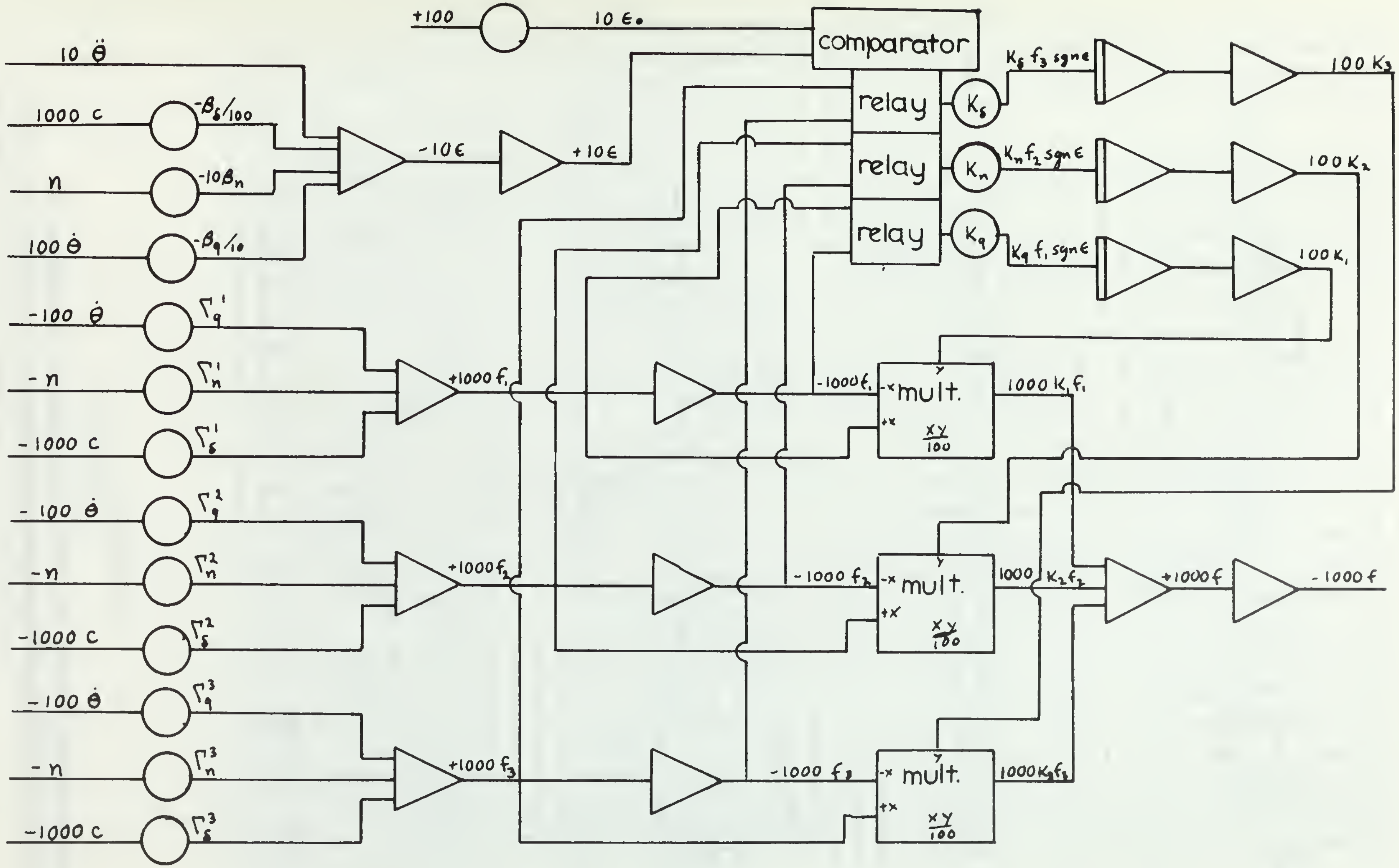

FIGURE 3 FEEDBACK SYSTEM ANALOG

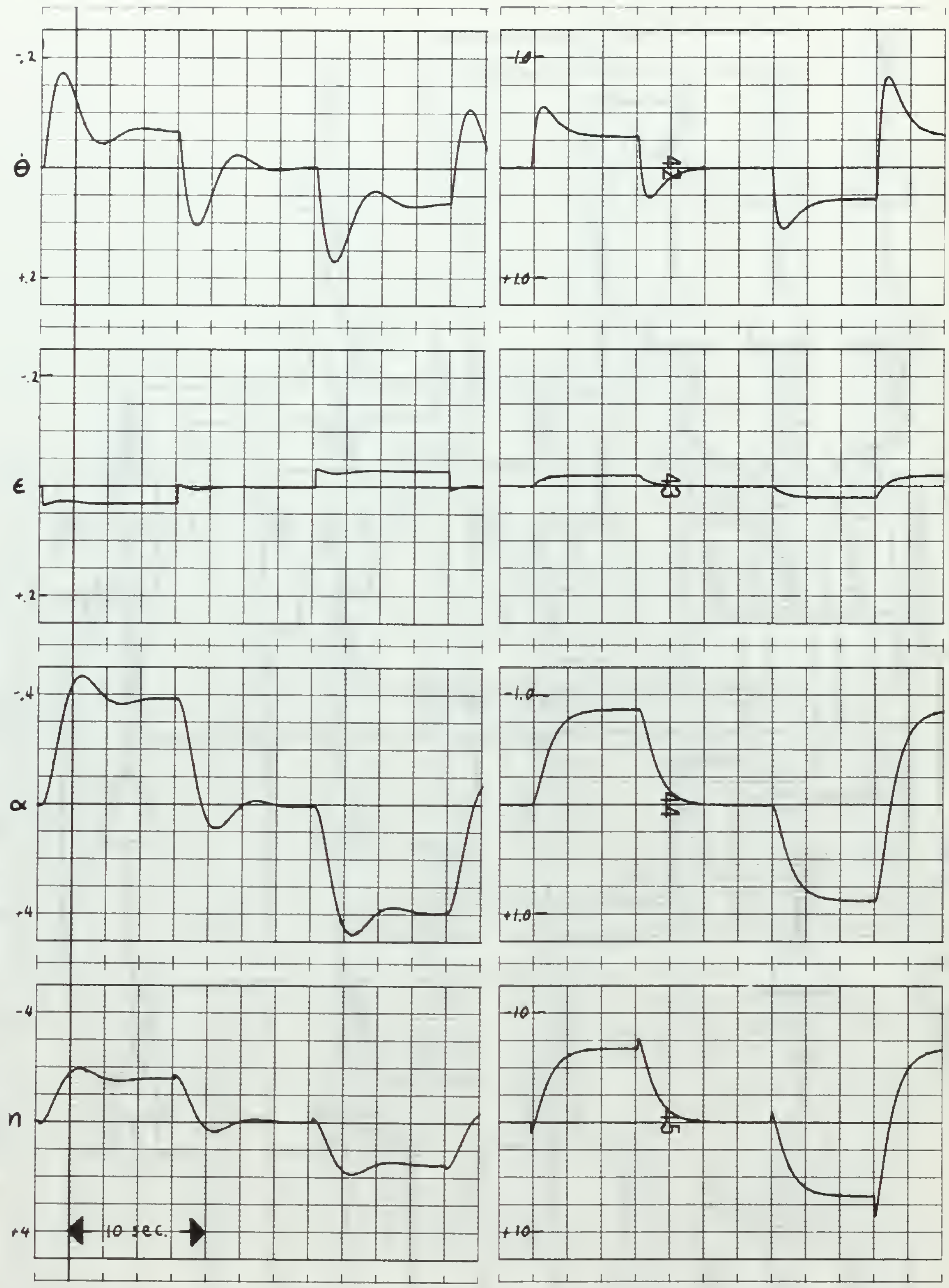

FIGURE 4

Plots for flight condition 0002, showing the response of the basic aircraft alone and with feedback.

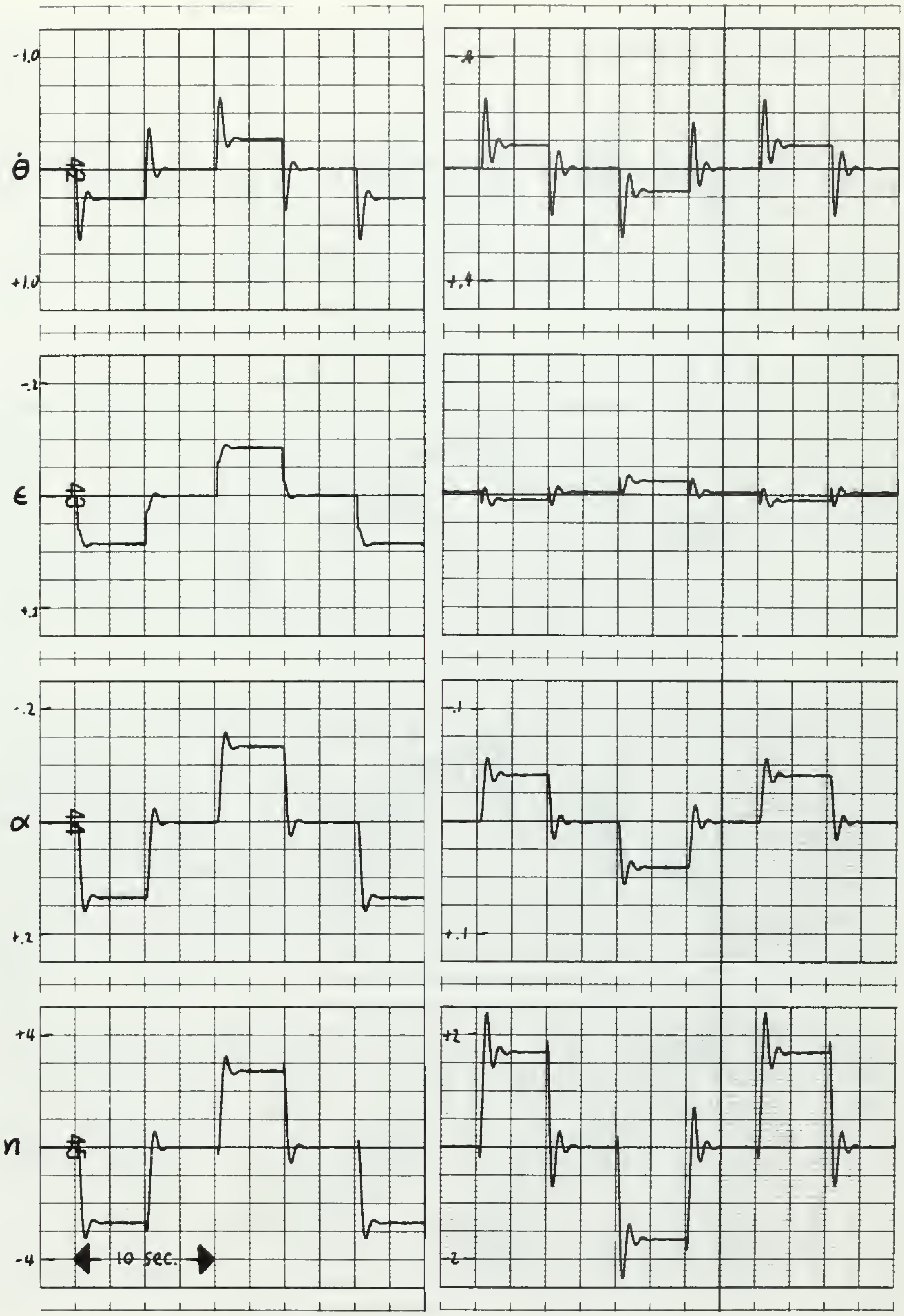

FIGURE 5

Plots for flight condition 0009, showing the response of the basic aircraft alone and with feedback.

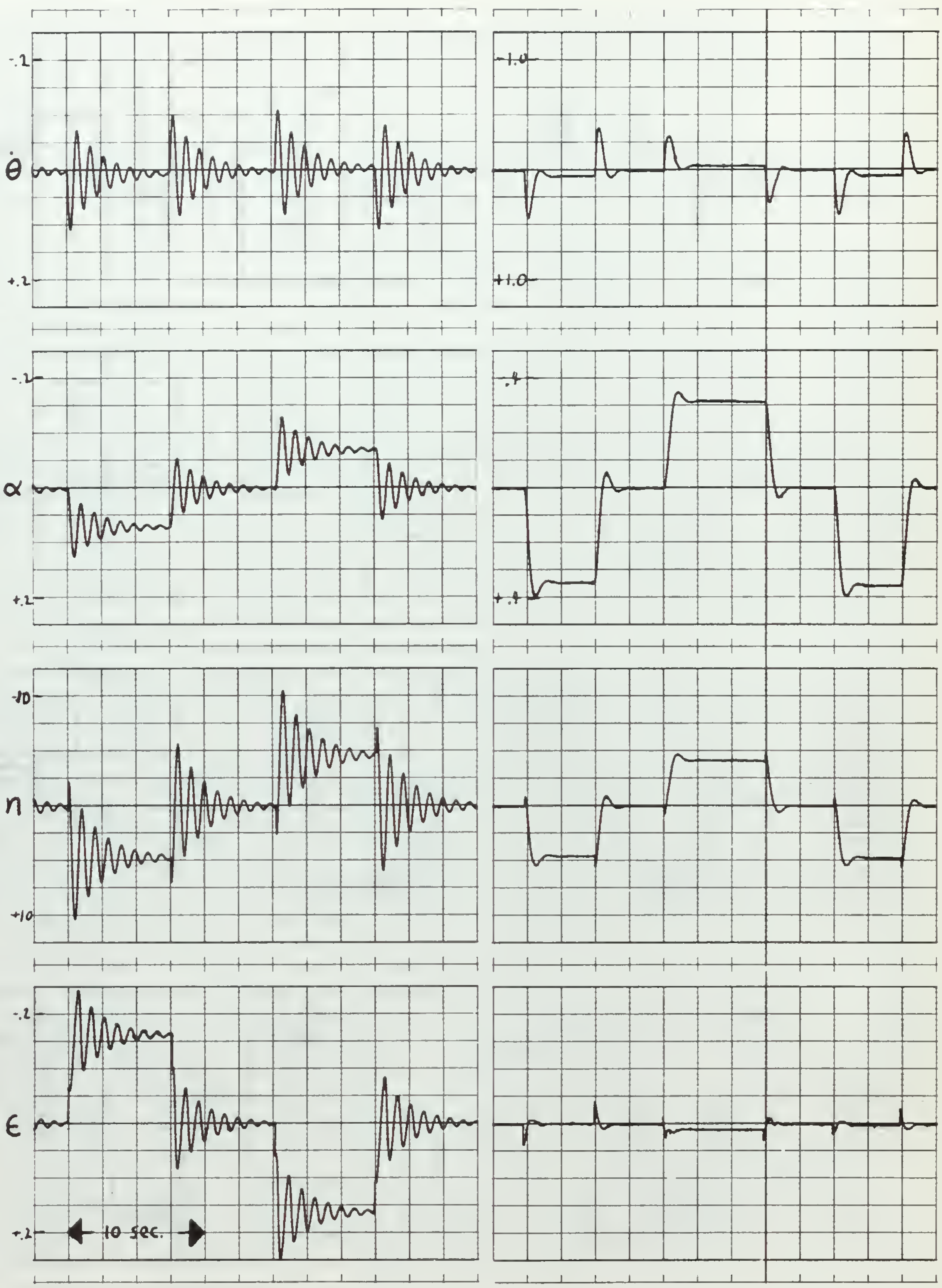

FIGURE 6

Plots for flight condition 5020, showing the response of the basic aircraft alone and with feedback.

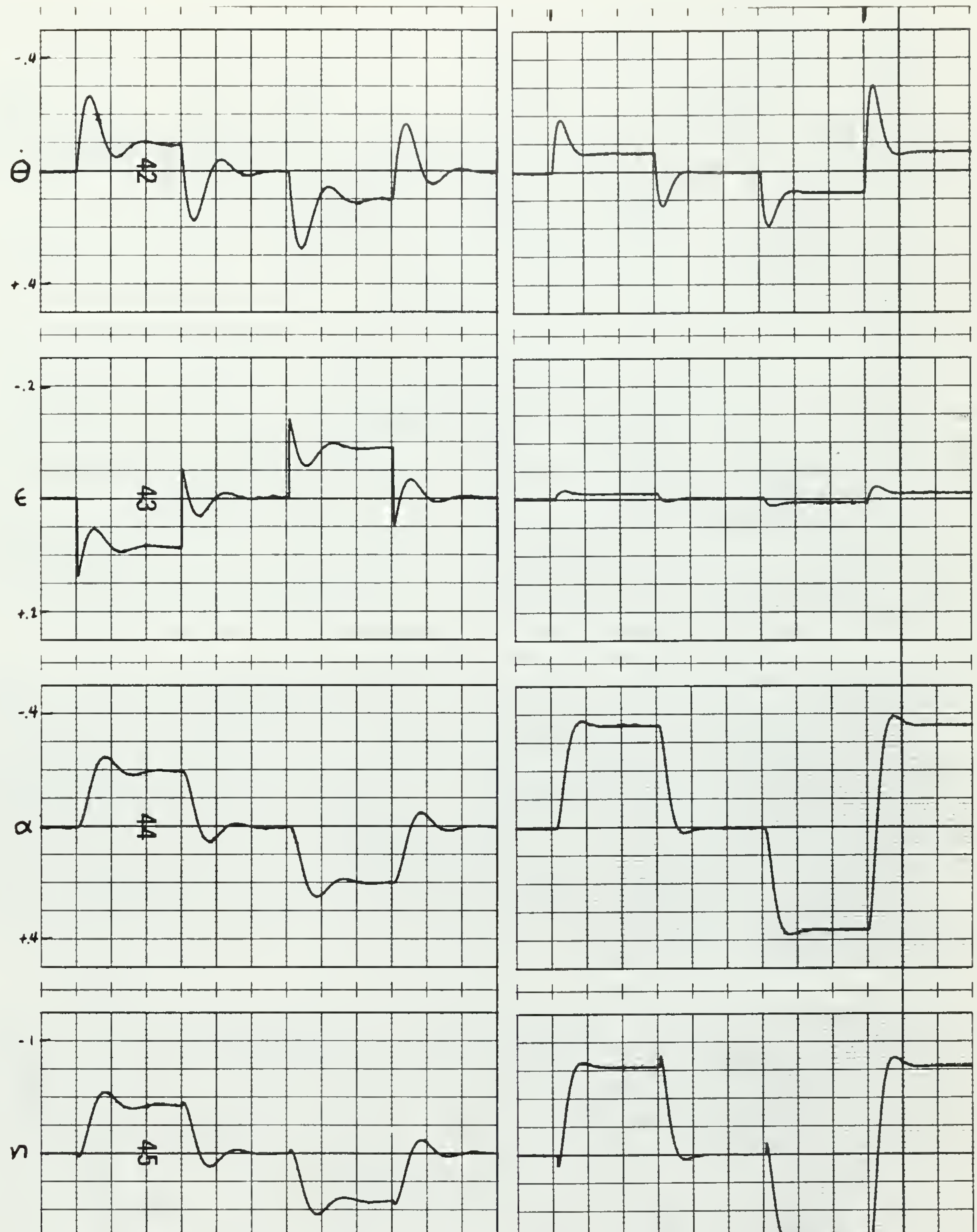

FIGURE 7

Plots for flight condition 1504, showing the response of the basic aircraft alone and with feedback.

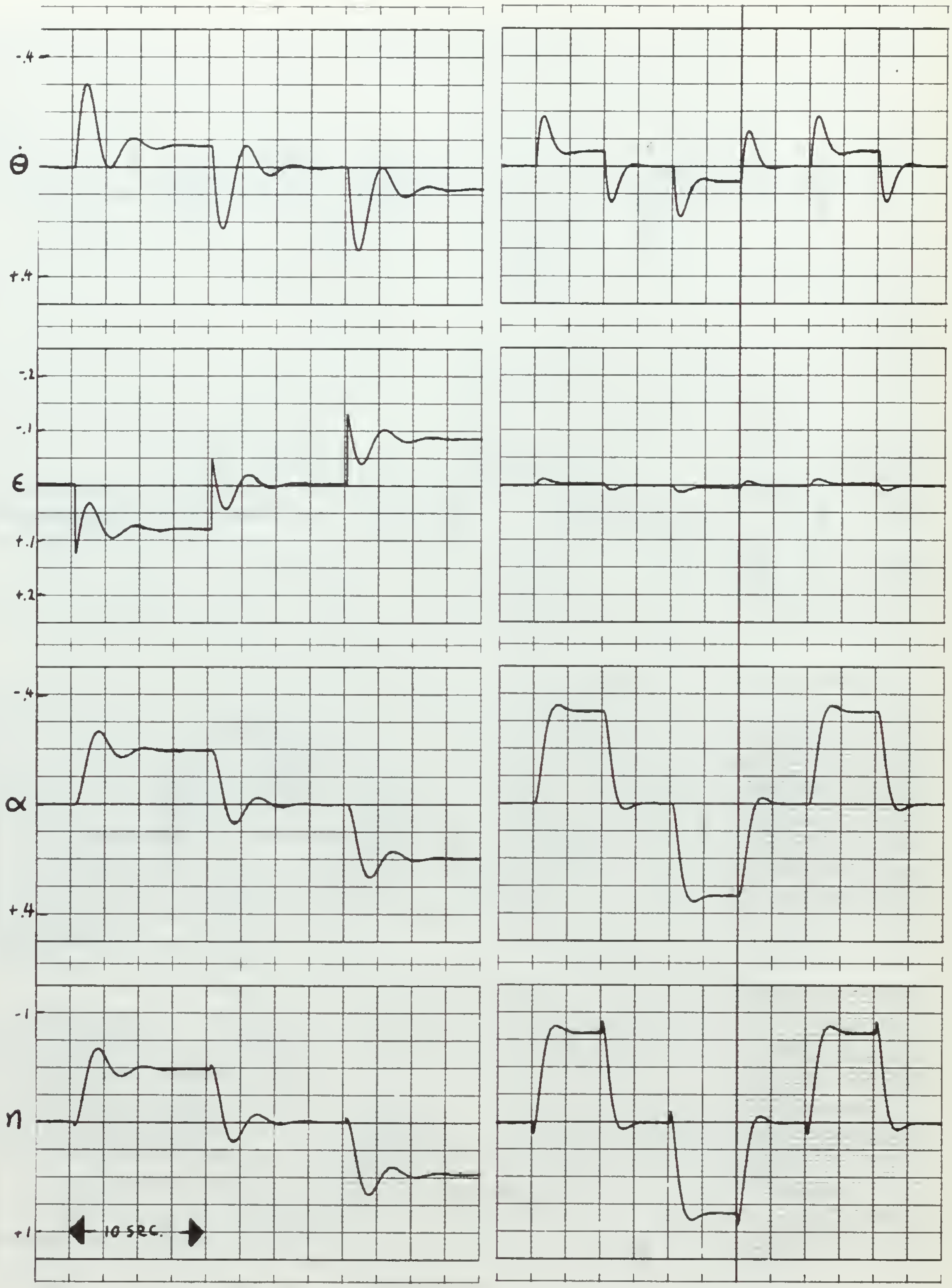

FIGURE 8

Plots for flight condition 3006, showing the response of the basic aircraft alone and with feedback.

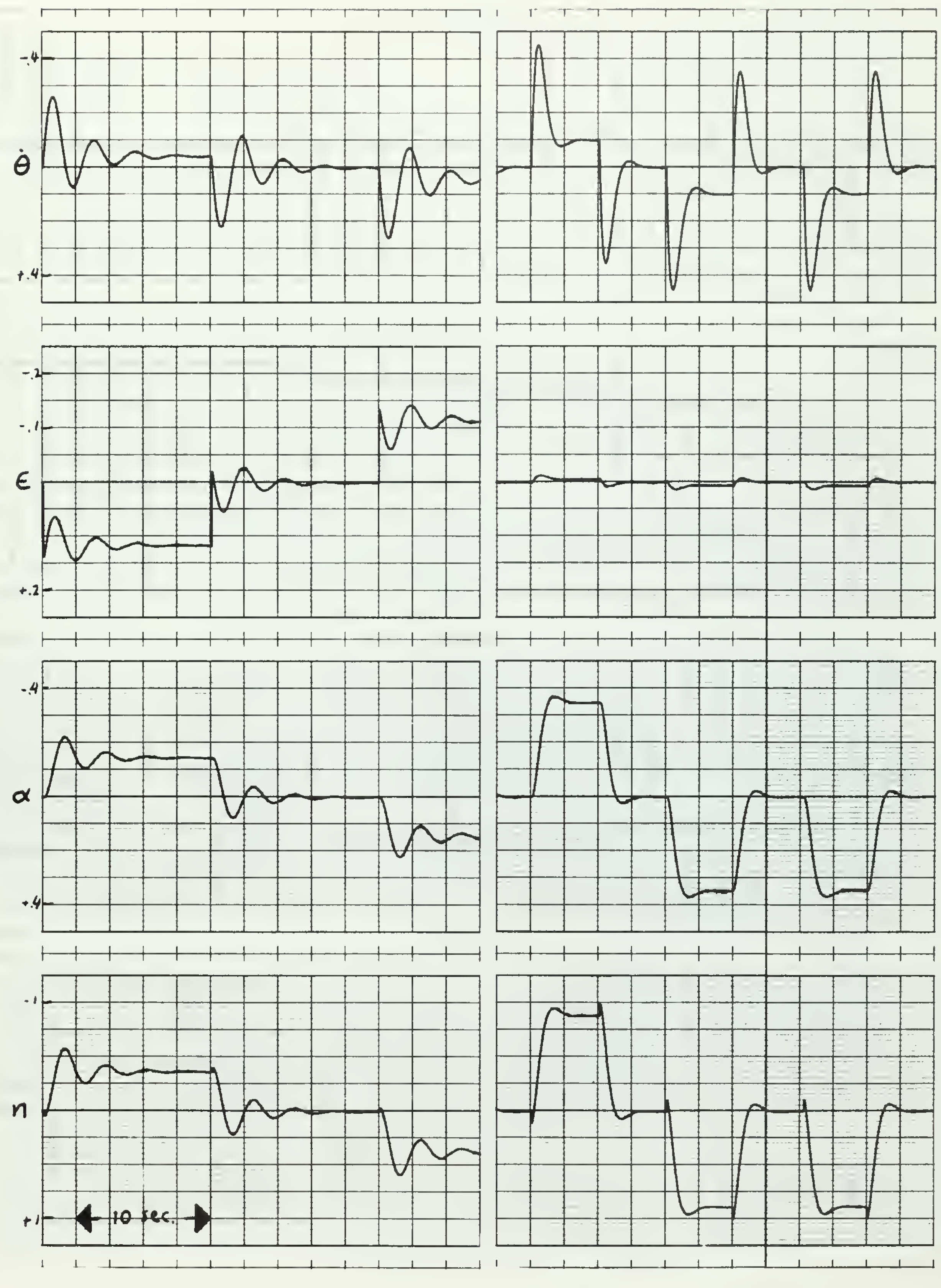

FIGURE 9

Plots for flight condition 4509, showing the response of the basic aircraft alone and with feedback.

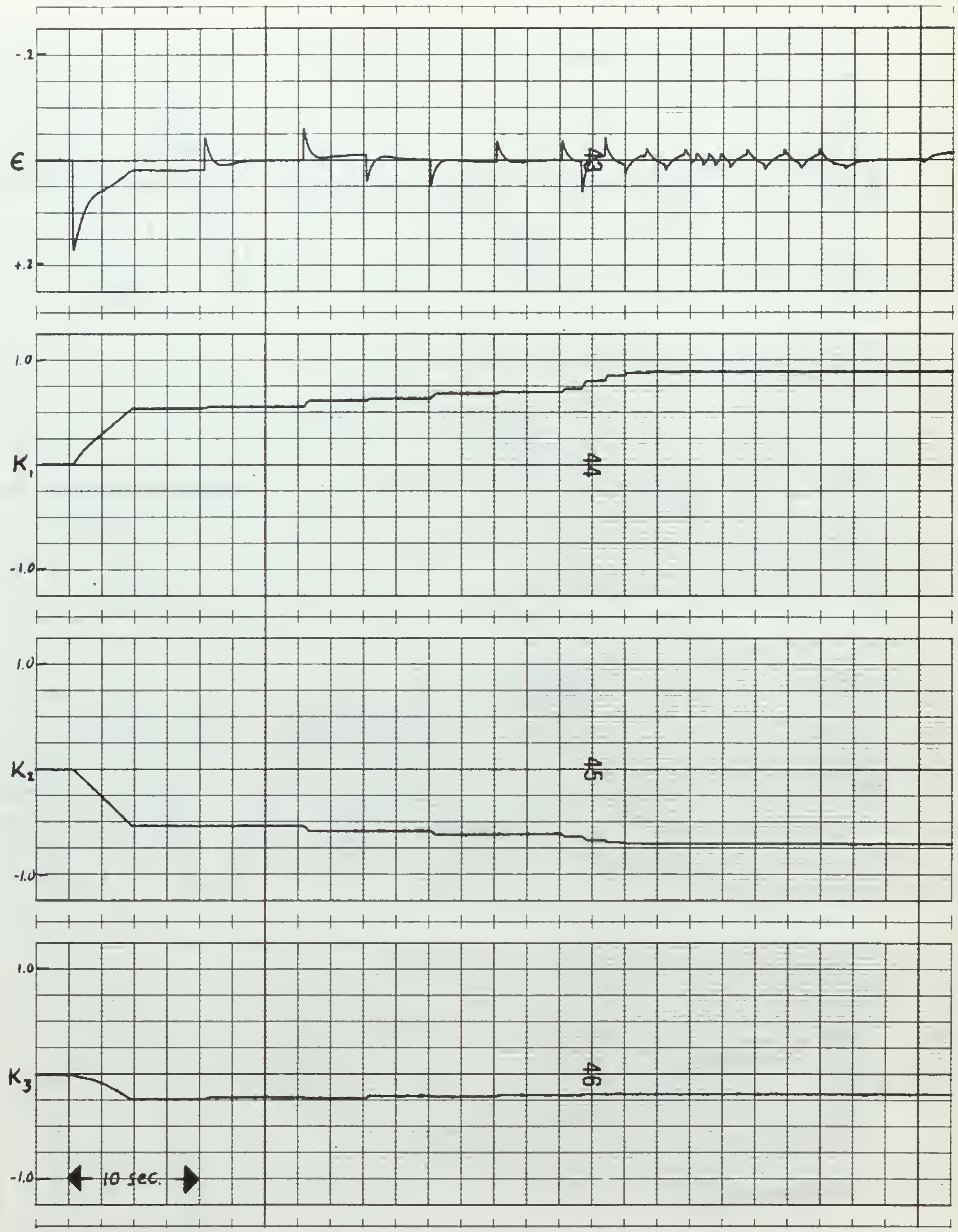

FIGURE 10

Plot for flight condition 0002, showing K's driving toward their steady state values.

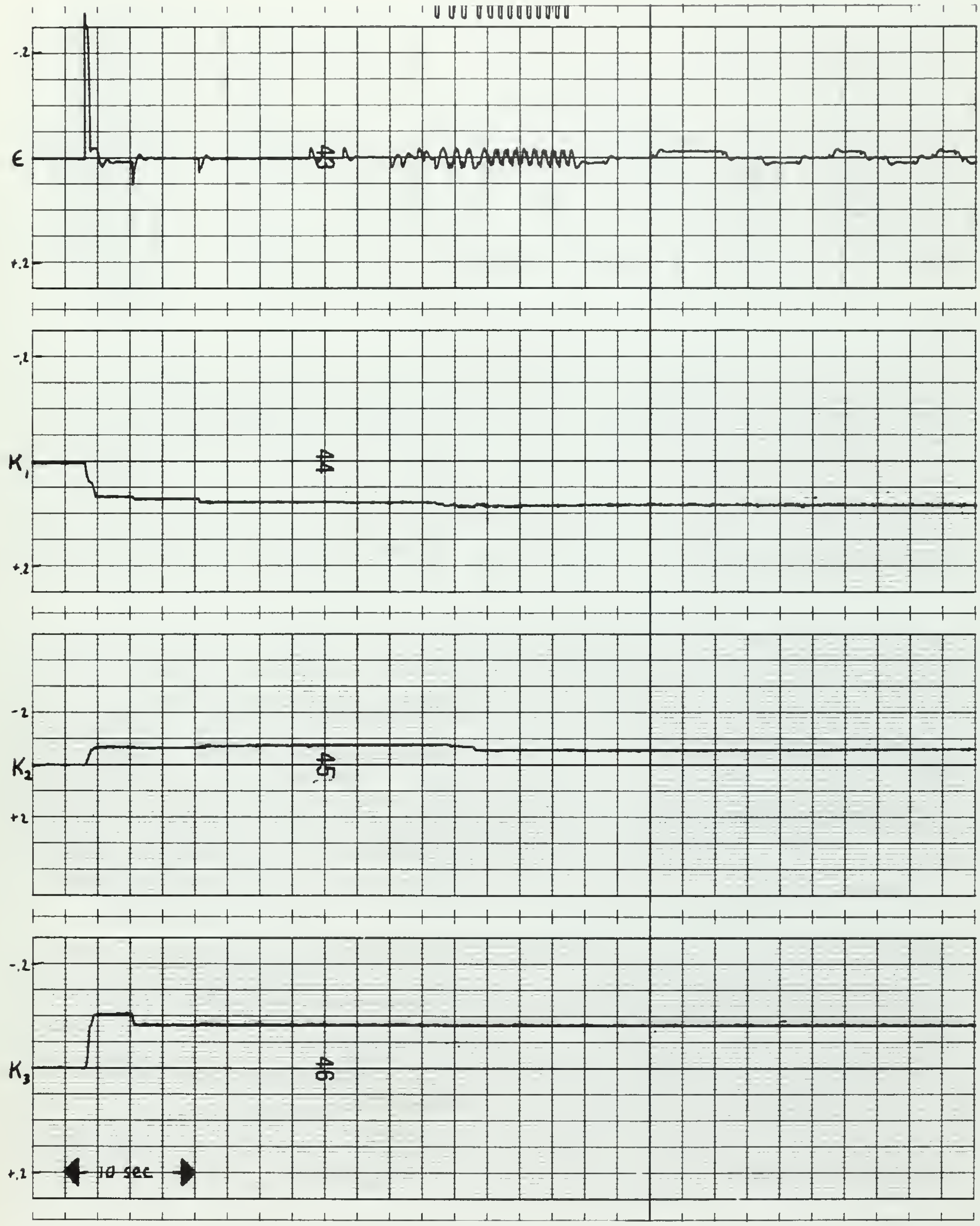

FIGURE 11

Plot for flight condition 0009, showing K's driving toward their steady state values.

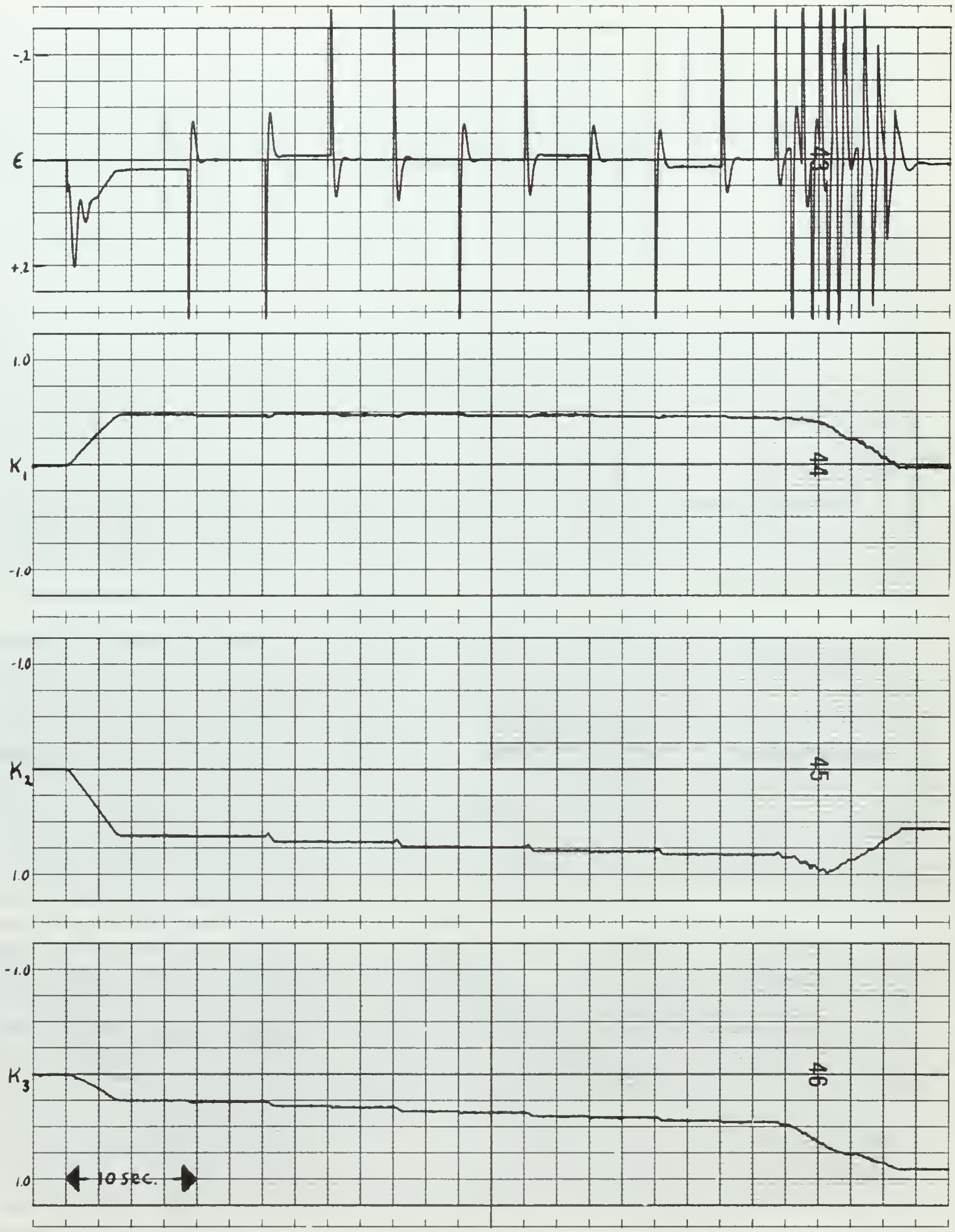

FIGURE 12

Plot for flight condition 5020, showing K's driving toward their steady state values.

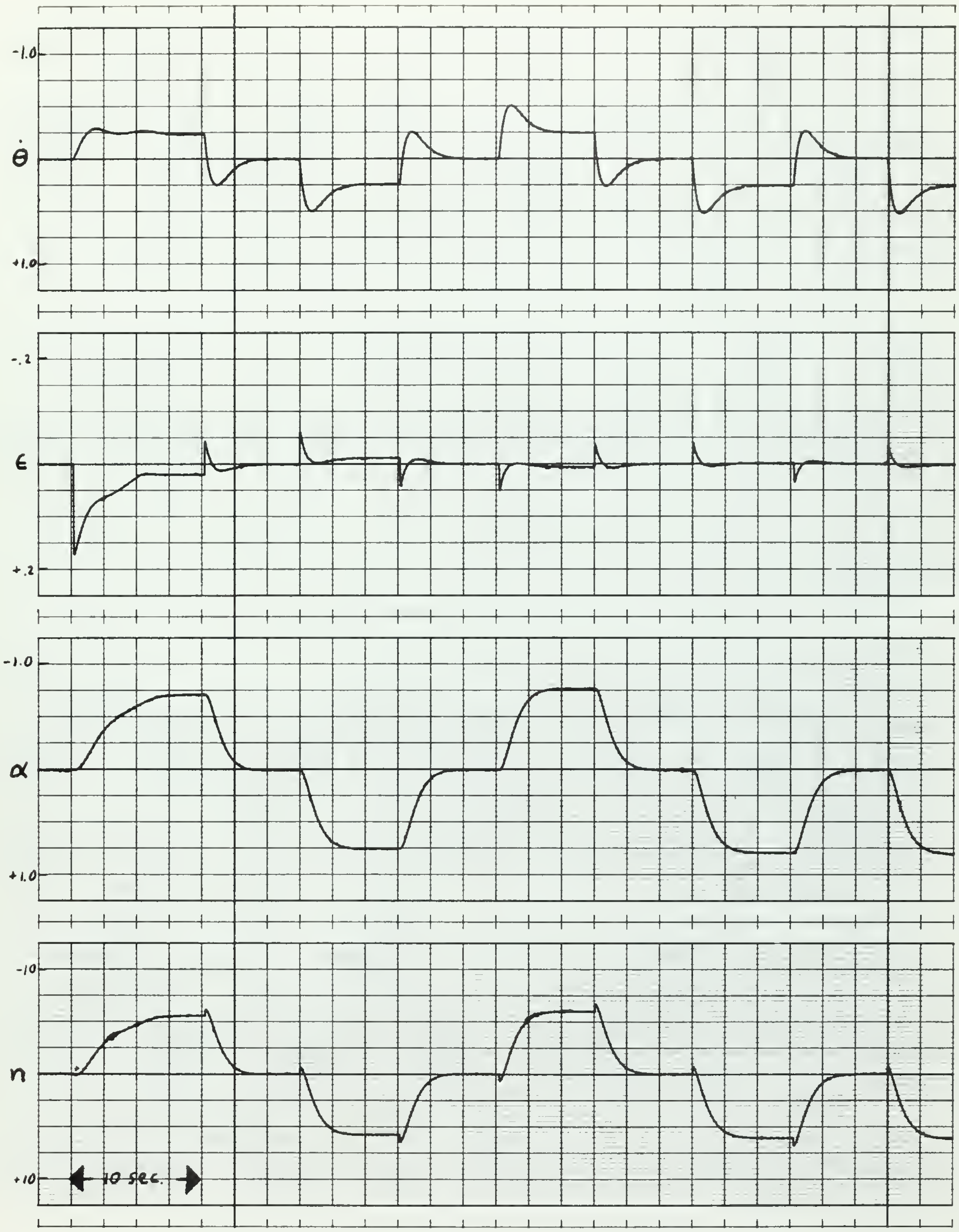

FIGURE 13

Plot for flight condition 0002, showing the change in response as the K's go from zero toward their steady state value.

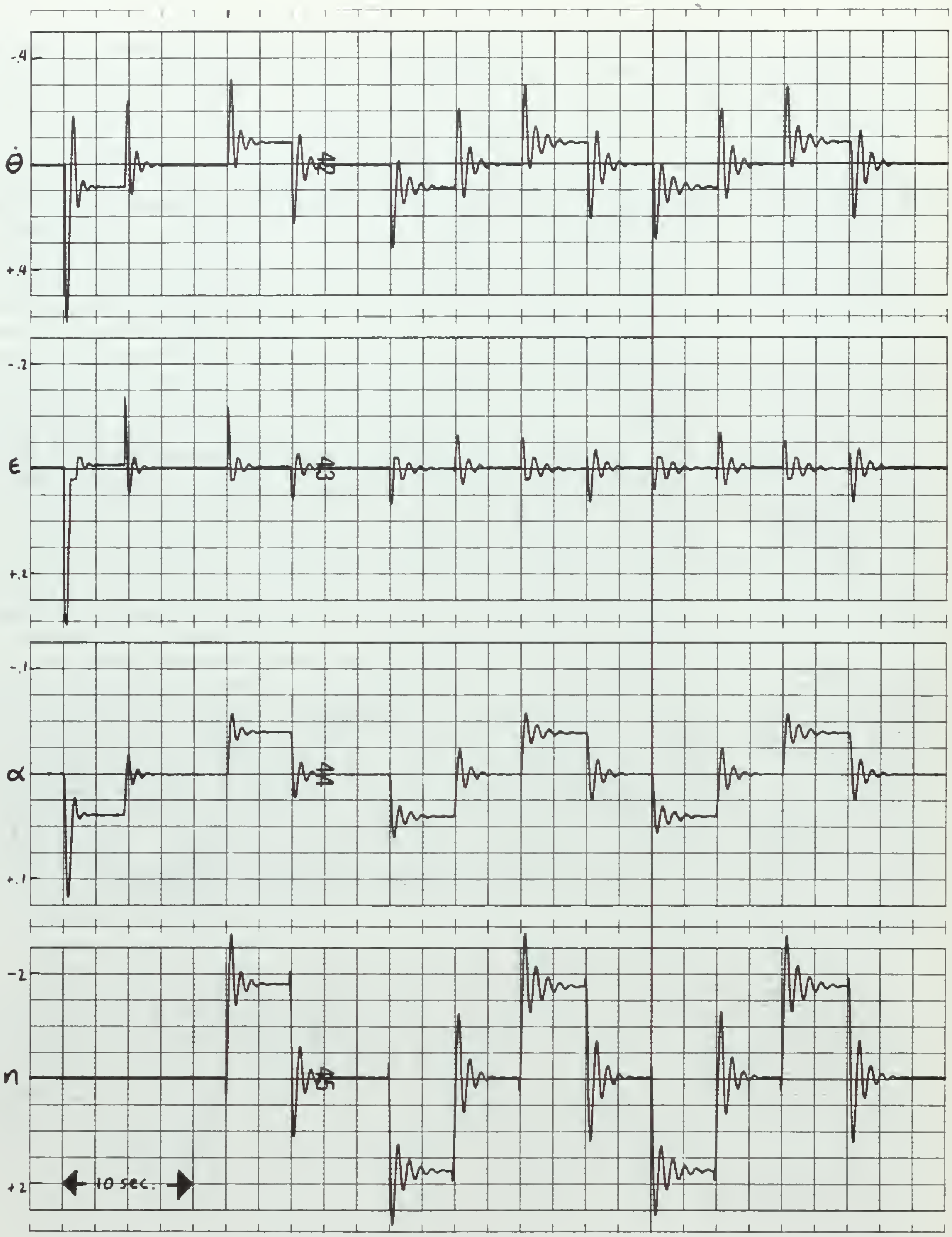

FIGURE 14

Plot for flight condition 0009, showing the change in response as the K's go from zero toward their steady state value.

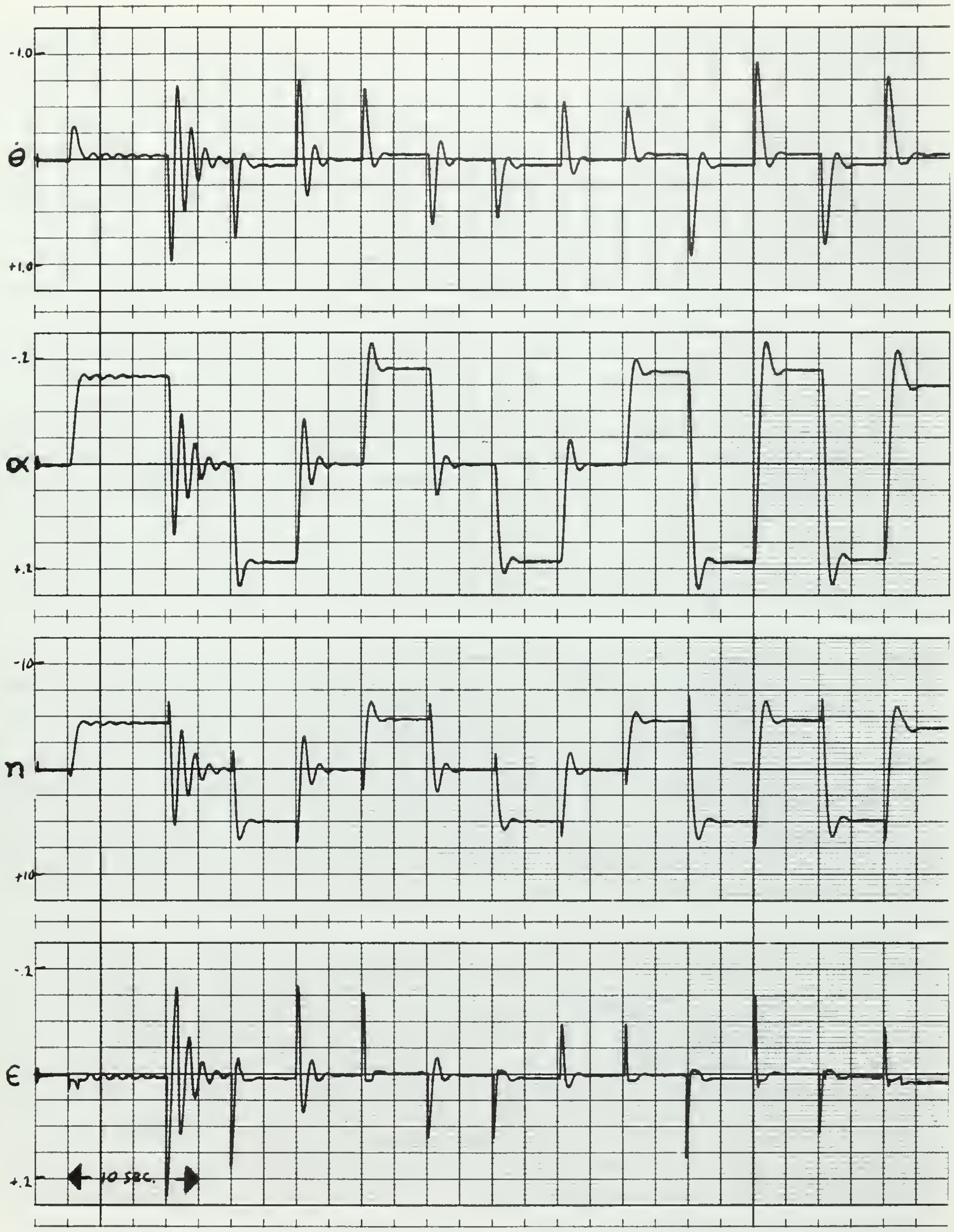

FIGURE 15

Plot for flight condition 5020, showing the change in response as the K's go from zero toward their steady state value.

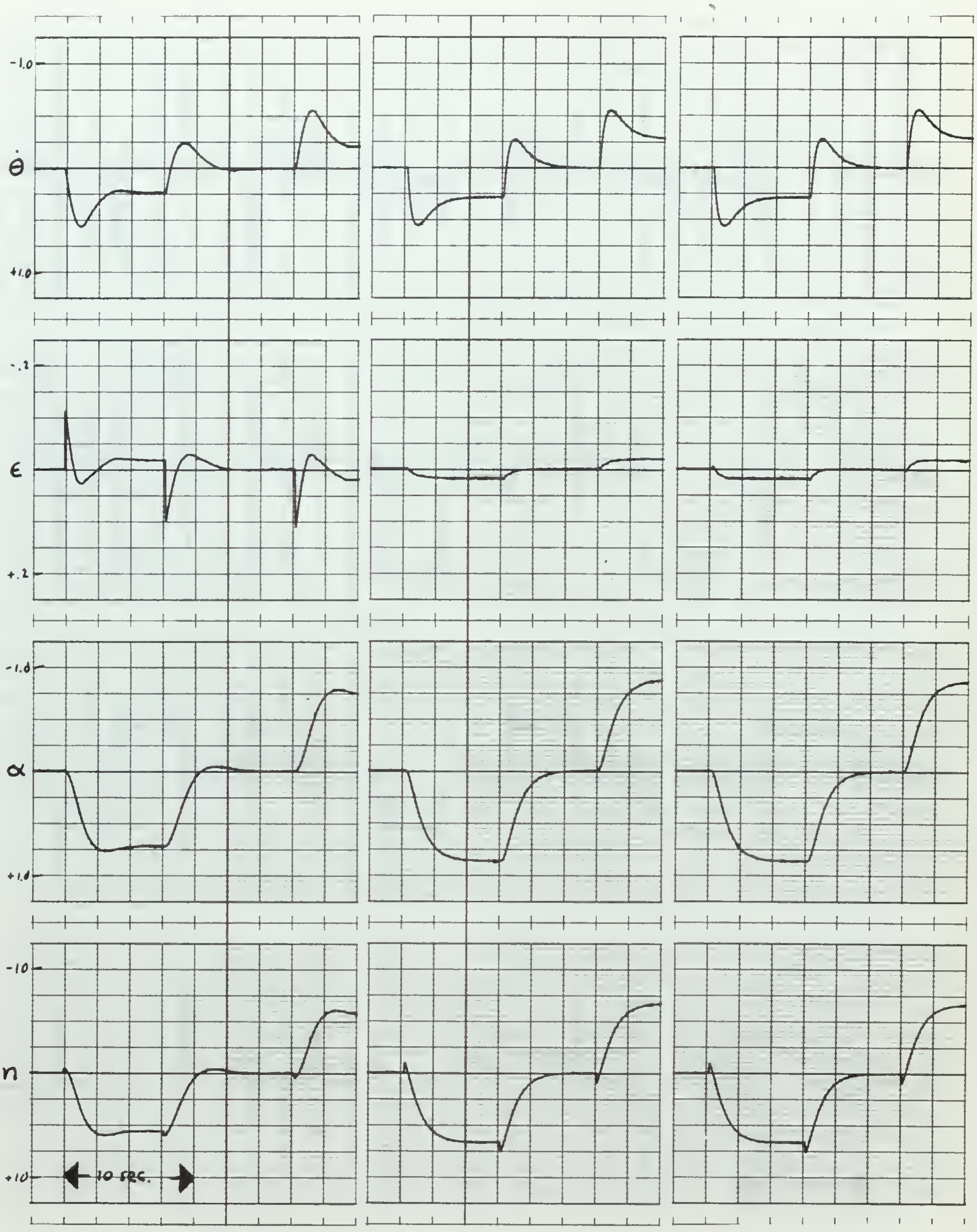

FIGURE 16

Plots for flight condition 0002, with the value of Γ_q^1, Γ_q^2, and Γ_q^3, respectively, set to zero.

FIGURE 17

Plots for flight condition 0002, with the value of Γ_n^1 , Γ_n^2 , and Γ_n^3 , respectively, set to zero.

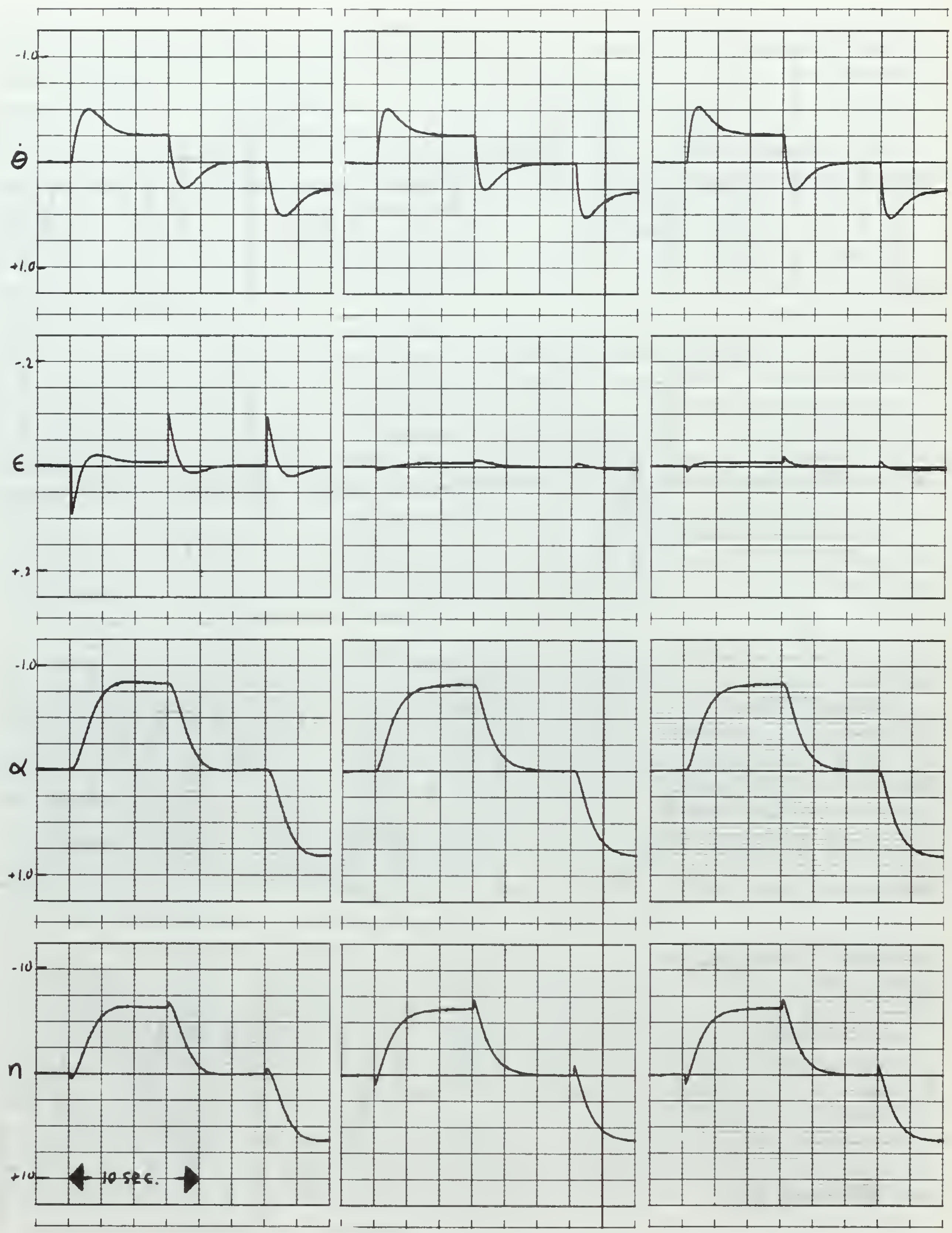

FIGURE 18

Plots for flight condition 0002, with the value of Γ_δ^1, Γ_δ^2, and Γ_δ^3, respectively, set to zero.

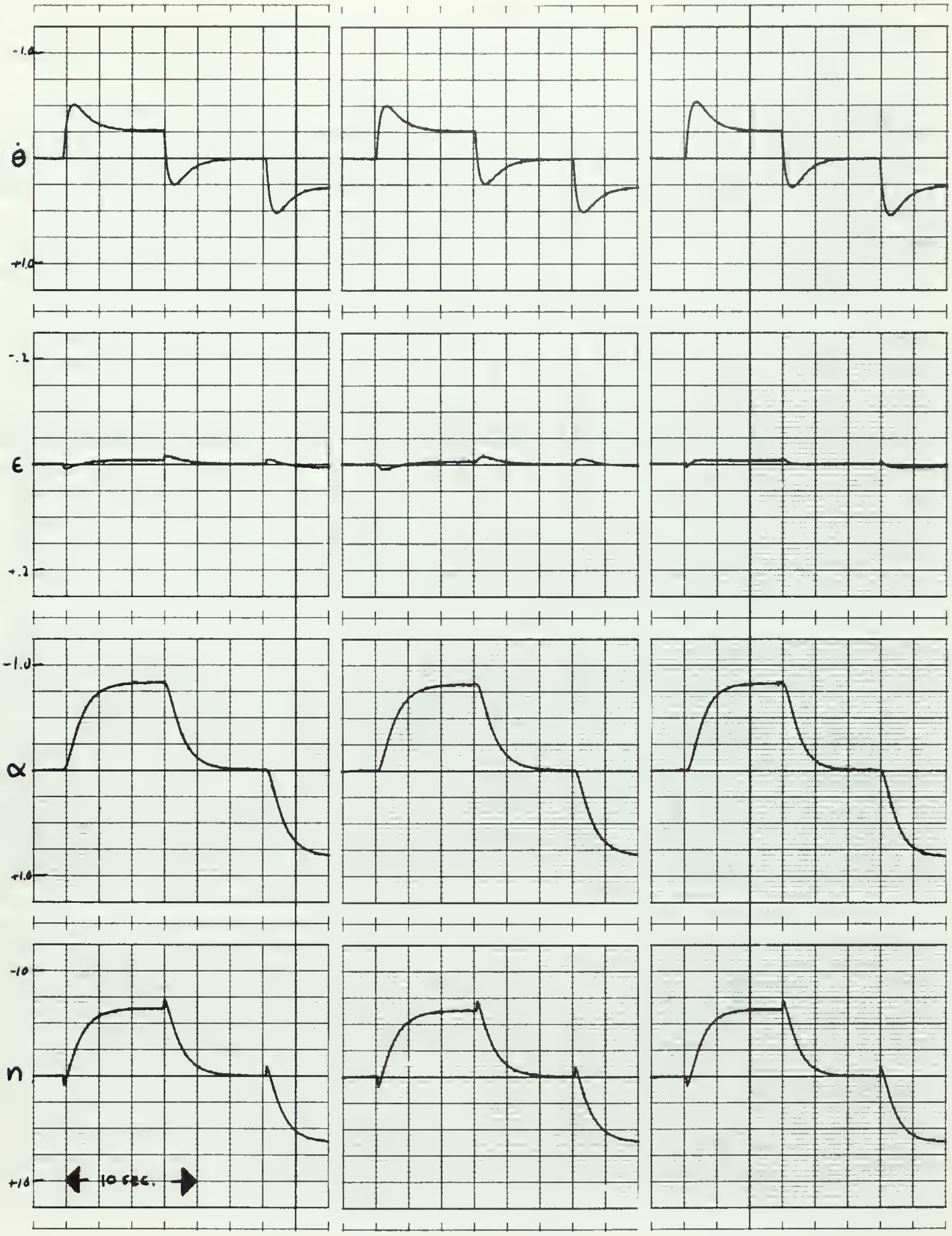

FIGURE 19

Plots for flight condition 0002, with the value of Γ_q^1 , Γ_q^2 , and Γ_q^3 , respectively, set to one.

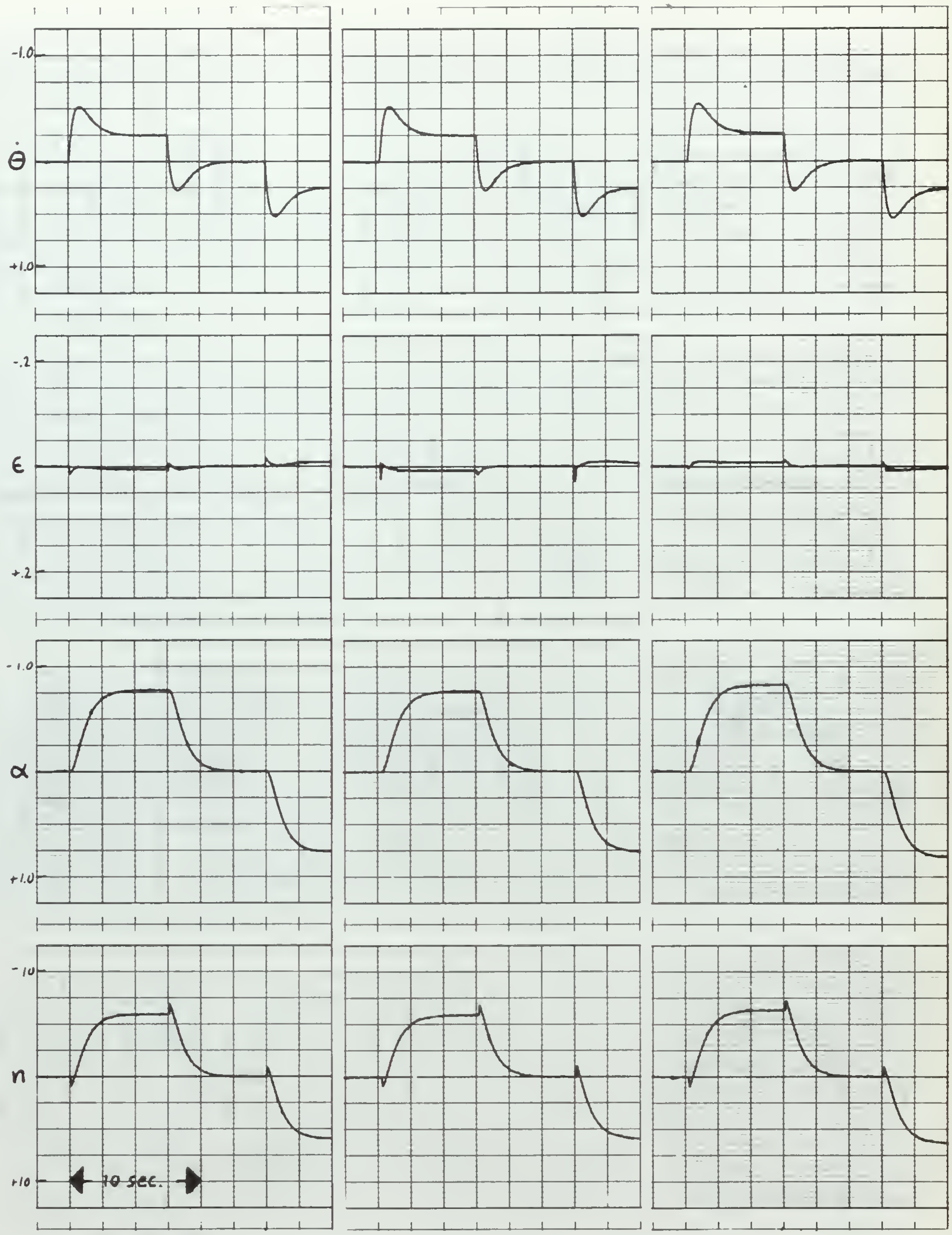

FIGURE 20

Plots for flight condition 0002, with the value of Γ_n^1 , Γ_n^2 , and Γ_n^3 , respectively, set to one.

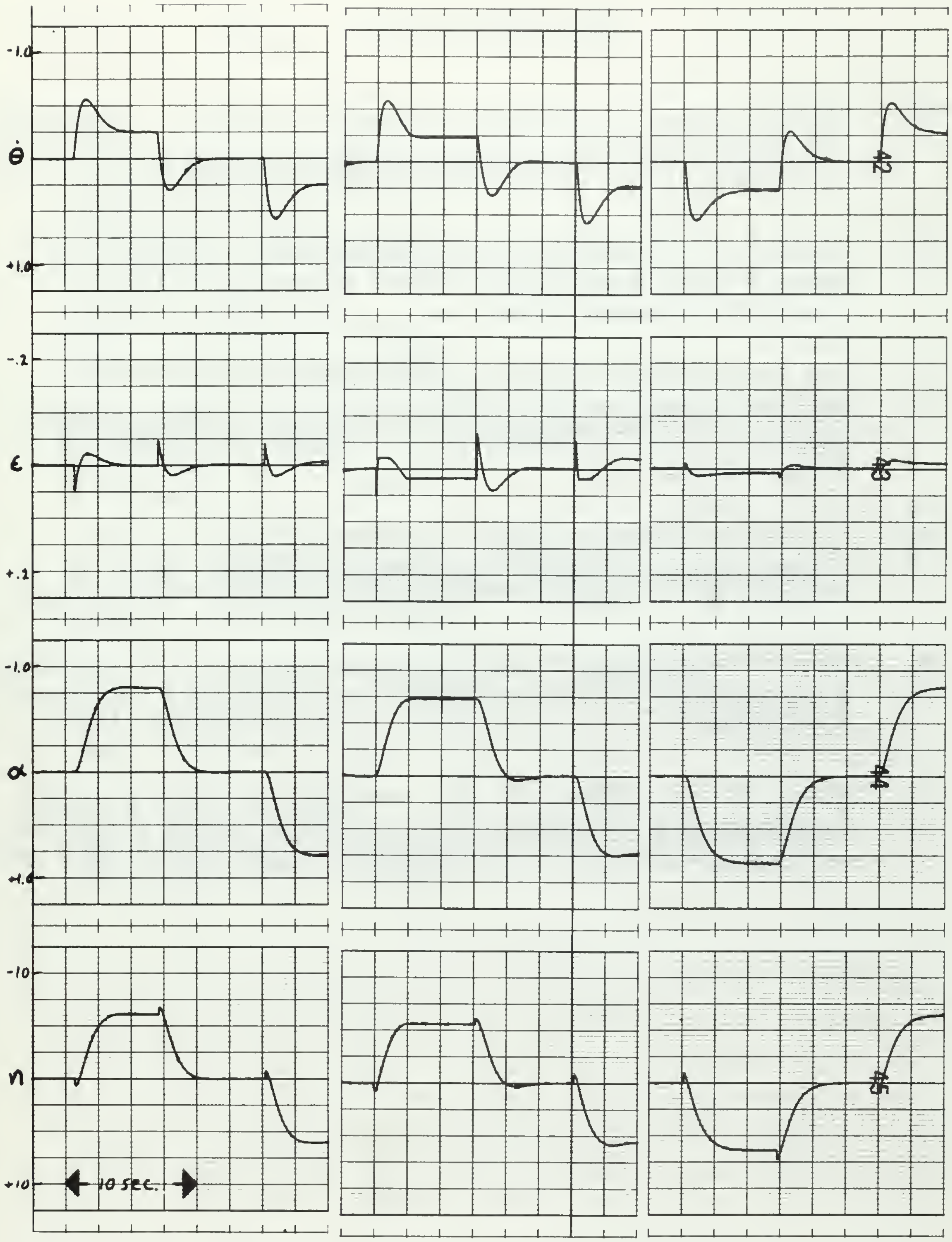

FIGURE 21

Plots for flight condition 00.02, with the value of Γ_s^1, Γ_s^2, and Γ_s^3, respectively, set to one.

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13. ABSTRACT

An adaptive control scheme may provide the best approach to the problem of accommodating automatic flight control systems to the variations of dynamic characteristics encountered over the flight envelope of the aircraft. The C*-Criterion for aircraft time response provides a basis for the design of such a system.

To provide a margin of safety, all control systems have redundant channels for emergency use. An adaptive control system can be designed to be self-organizing, and in such a configuration can provide its own failure monitoring, resulting in a simpler and more efficient system.

This study shows that such a system is feasible, that the response of the system is within the limits set forth in the C*-Criterion, and that the self-organizing characteristics provide reliable operation over the whole range of flight operations.

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